Demonstration of a Precision Data Reduction Technique for Navigation of the Galileo Spacecraft

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introduction

Navigation of interplanetary spacecraft is accomplished by fitting several different data types to a model of the forces affecting the spacecraft and error sources which affect the data. Several types of radiometric data are currently in use, such as two-way Doppler, two-way range and Delta Differential One-way Range (ΔDOR). Of these, Doppler data, which measures changes in range along the line-of-sight to the spacecraft, has been highly reliable and is general 1 y the cornerstone of deep space navigation systems. The precision of two-way Doppler data varies depending on the uplink and downlink frequencies, the distance to the spacecraft, and the signal-to-noise ratio, but it can be as good as 0.2 rends over 60 second measurement intervals. However, the data are rarely used at this level due to limitations in modeling various error sources and in computational capablilities. Recent] y, advances have. been made in modeling the principal Doppler measurement error sources, and with the advent of high speed workst at ions available to flight projects, t he possibilit y arises of improving the navigat ion performance of Doppler d at a. Error anal yses have shown that substantial improvements are theoretically possible, but the real validation comes from testing the proposed innovations in actual flights. This paper presents results from a demonstration of a new sequent ial filter model and data reduct ion technique, dubbed the "enhanced filter", performed with radio tracking data obtained from the Gal ileo spacecraft. Two cases are described, one from the Earth-2 f] yby of December 8,1992, and the other from tile approach phase of the asteroid Ida encounter, which occurred on August 28, 1993.

The Galileo Mission

The Galileo spacecraft was launched on October 18, 1989 towards Jupiter using a Venus-Earth-Earth Gravity Assist (VEEGA) trajectory. On December 8, 1992, the spacecraft flew by the Earth at an altitude of 303 km on the final gravity assist which propelled it to Jupiter. During the Earth-Jupiter cruise phase of the mission, the spat.ccraft encountered tile asteroid Ida. For each of these events, cst imated of the spacecraft trajectory were needed to design the targeting maneuvers.

Due to the failure of the High Gain Antenna (HGA) to deploy, the spacecraft navigation was accomplished using the Low Gain Antenna (LGA). The data from the LGA consisted of two-way Doppler, two-way range, and ΔDOR data points acquired at S-band frequencies. The ΔDOR data augments the Doppler and range by providing information in the directions perpendicular to the line-of-sight to the spacecraft. The operat ional trajectory solutions which used ADOR data provided one basis for comparison with the results obtained using the enhanced filler mode.1.

Filter Strategy

Orbit determination accuracy is heavily dependent on the models used to represent the spacecraft's orbit and the error sources affecting the data. The force models used to integrate the trajectory include the gravitational attraction of the sun, moon and the nine planets, solar radiation pressure, and spacecraft propulsive events. Error sources affecting the data include path length delays due to the troposphere anti ionosphere and crews associated with DSN station locations. Operationally, estimated parameters (parameters which were adjusted to obtain the best least-squares fit to the data) included the state, specular and diffuse values for solar radiation pressure, and three Cartesian velocity components of all propulsive events. Considered parameters (parameters which contribute to the formal uncertainty of the fit but are not actually adjusted) included tropospheric and ionospheric path delays, and station location uncertanties.

For the enhanced filter, all the consider parameters mentioned above were placed in the estimate list. In addition, four other parameters, three for the orientation of the earth and one representing biases in the Doppler caused by solar plasma, were also estimated. Except for the station locations, all were modelled as exponentially correlated process noise with zero mean. A batch-sequential filter algorithm was used to perform the fit.

Results

The first test case was for the Earth-2 encounter and used a data arc starting on October 15, 1992 and ending on November 22, 1992. The Doppler weight (assumed measurement uncertainty) used with the enhanced filter was 0.3 mm/s over 60 second intervals. Range data was also fitted using a weight of 1 km. The results are shown in Figure 1 in the Earth B-plane. As a comparison, a solution using a standard filter with Doppler weighted at 2 mm/s and range at 1 km is also shown on the figure. It can be seen on the figure that the enhanced Doppler solution came closer to determining the actual trajectory of the spacecraft as computed by the post flyby reconstruction (also shown on the figure) than the standard Doppler solution. Additionally, the enhanced filter results in smaller uncertainties.

The second test case used a data arc which started on February 10, 1993 and ended on April 27, 1993. The solution was mapped to the Ida B-plane. Once again, the Doppler data for the enhanced filter were weighted at 0.3 mm/s. The range. data for this case was weighted at 100 m. This solution (shown in Figure 2) is compared with two others; the first using standard Doppler and range data only, and the second using Doppler, range and ADOR data. It can be seen from the figure that both the enhanced filter solution and the solution using ADOR data agree very well with each other with comparable uncertainties. Both were also consistent with the standard Doppler and range only solution, but the latter had a much larger error ellipse.

These test cases provide the first verification of the enhanced filter model in an operational mission. in particular, the second case shows that the results obtained by this method are at least comparable to that provided with ADOR data, without the cost and complexity associated with the ADOR system.

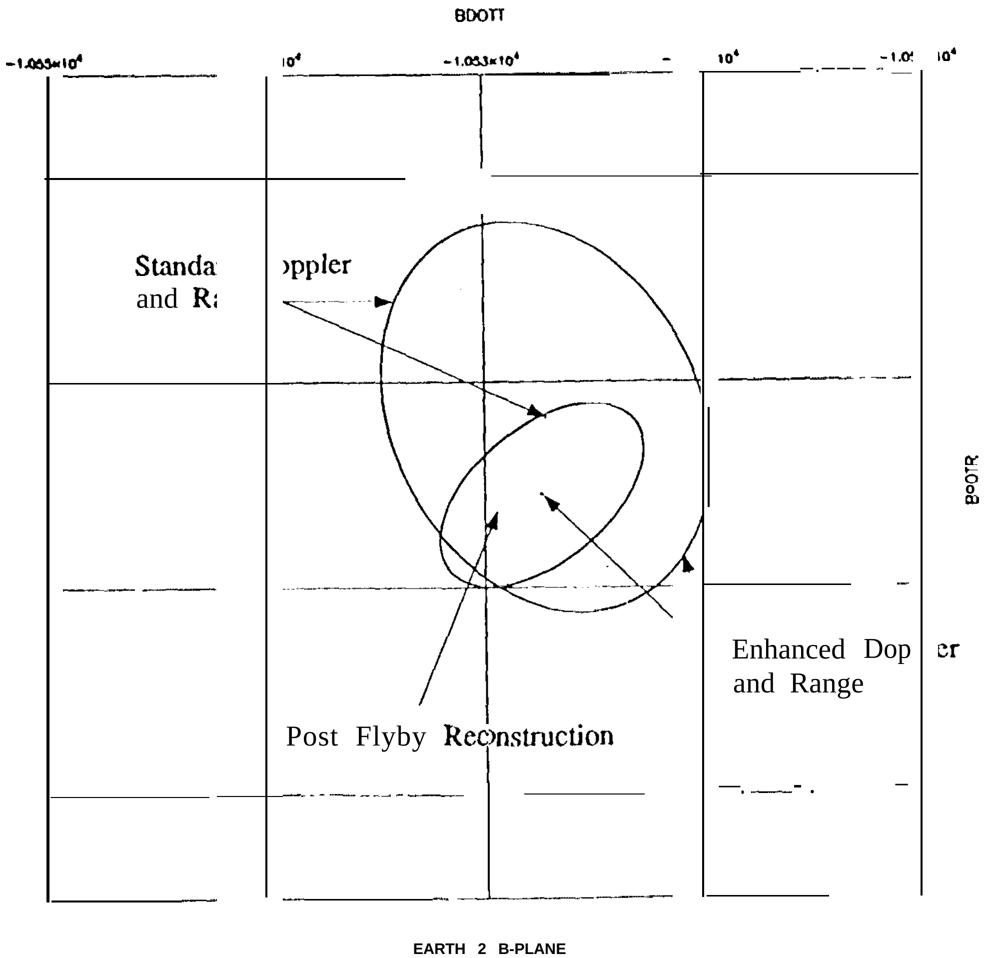

Figure 1: Comparison of Enhanced Doppler and Standard Doppler Solutions

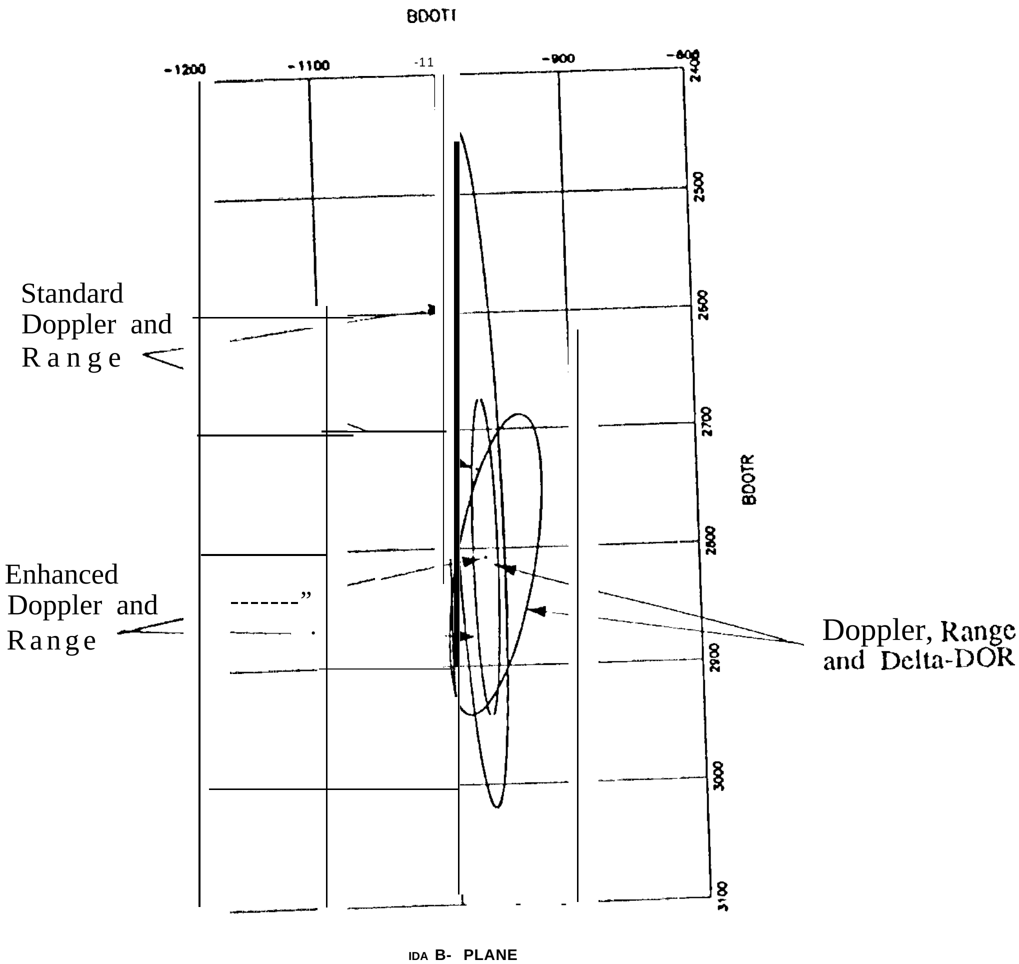

Figure 2: Comparison of Standard Doppler, Enhanced Doppler. and Delta-DOR Solutions in the Ida B-plane

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GALILEO ORBIT DETERMINATION FOR THE EARTH-2 ENCOUNTER

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On December 8, 1992, the Galileo spacecraft ftcw past the Earth for the second time, completing its Venus-Earth-Earth gravity assist trajectory and placing it on a course for Jupiter. This paper discusses the orbit determination (OD) procedure leading up to the Earth-2 encounter. The OD task involves fitting S-Band doppler, range and ΔDOR tracking data to a mathematical model of the trajectory, and then projecting that model forward to the encounter time. Parameters estimated include. the state. at epoch, solar radiation effects, and effects from the onhoard attitude control subsystem. The information obtained was used to design a series of four trajectory correction maneuvers which adjusted the. orbit to achieve the desired flyby conditions. The paper concludes with a summary and discussion of the key orbit determination solutions, showing a steady improvement in OD acuracy as the spacecraft neared encounter,

INTRODUCTION

The Galileo spacecraft was launched on October 18, 1989 to begin a 6-year voyage to Jupiter. Due to the low injection AV available from the Inertial Upper Stage (l[JS), Galileo had to usc a series of gravity assists to obtain enough energy to reach Jupiter. The gravity assists were provided by onc flyby of Venus and two flyby's of Earth and was called the VEEGA (Venus-Earth-Earth -Gravity-Assist) trajectory [Ref. 11. On December 8, 1992, the Galileo spacecraft flew by the Earth for the *second* and final Lime to complete the VEEGA trajectory which sent the spacecraft on its way to Jupiter.

The VEEGA trajectory was designed to minimize the propellant needed to get the spacecraft to Jupiter. Successful completion of the VEEGA trajectory requires that the spacecraft be kept relatively close to the designed trajectory, Orbit determination (OD) involves using several types of tracking data to compute the actual trajectory of the spacecraft and compare it to the nominal one. This information can then be used to design maneuvers to achieve the desired targets at the planetary encounters. For the Earth-2 fly-by, the targeted aimpoint was at an altitude of 304 km over the south Atlantic ocean. A miss of only a fcw kilometers would have required the use of additional propellant to correct the flyby errors, This stringent targeting requirement and the c1ow proximity of the flyby to the Earth required precise navigation. In addition, the navigation had to be performed using only the Low Gain Antenna (LGA) due to the failure of the High Gain Antenna (HGA) to deploy. This paper details the OD process used to navigate Galileo through the second Earth flyby.

DATA

The only means by which a spacecraft's actual trajectory can be determined is by the usc of tracking data. 'lb-cc data types were employed for OD during the Earth-2 encounter: doppler, range, and ADOR (Delta Differential One-way Range). Because the LGA was wed, these data types were at S-band frequencies rather than X-band (which is available only from the HGA). Doppler data measures the frequency shift caused by the line-of-sight component of the velocity of the spacecraft with respect to the tracking station. Errors in this data type arc caused by delays induced by the medium in which the signal travels, specifically by the ionosphere and troposphere of the Earth. These errors arc generally removed by calibrating the data with external information of the media delays, but residual noise still remains. Since S-band doppler data is affected more by ionospheric and tropospheric delays than X-band doppler, its noise level is generally an order of magnitude higher. For this phase of the mission, however, the doppler data was quite good with the noise always under 1 mm/see for a 60-second count time. Nevertheless, the doppler data was nominally weighted at 2 mm/s to account for model deficiencies in the filter. Onc additional point to note about the doppler data is that the LGA is mounted such that it points along the spin axis of the spacecraft. The spinning of the antenna, because it is in motion relative to the tracking station, imparts a doppler shift to the signal. This shift, or bias, sums linearly to the doppler shift caused by the translational velocity of the spacecraft with respect to the tracking station. In the spacecraft's usual, dual-spin configuration, a 109.5 mHz bias is imparted by the spin [Ref. 2].

Range data is acquired by measuring the time between the transmission and reception at Earth of a ranging signal. Since the point-to-point accuracy of the range is dependent on signal strength, the range data suffered more than Doppler data from the unavailability of the I] GA. Duc to the distance of the spacecraft from the Earth and the resulting signal-to-noise rat io, range was unavailable before June, 1992. From June through early August, sporadic passes were obtained with the root-mean-square (rms) of the noise hovering near a kilometer. Subsequently, range data was consistently available, and the point-to-point noise quickly decreased until it was about 1 m around the encounter. However, station calibration errors and solar plasma effects tended to bias entire range passes by approximately 5-10 m. Operational y, the range data was assigned an uncertainty of 100 m. This accuracy, although much larger than its theoretical limit, was considered adequate because higher accuracies *were* not necessary for the encounter. Moreover, it was of the same order of magnitude as the uncertainties in the other data and trajectory models used in the estimation process. Therefore, deweighting the range to this level was one way to prevent the introduction of systematic errors into the OD.

ADOR data is an interferometric data type in which one-way range data from the spacecraft received at two stations is differenced with similar signals from a nearby quasar [Ref. 2]. This differencing effectively removes much of the atmospheric effects and gives a measurement of the angular separation of the spacecraft from the quasar (whose position is well known). The observable quantity in the ADOR measurement is the time delay of the quasar signal observed at the two stations subtracted by the time delay of the spacecraft signal at the same two stations. The measurement is usually given in terms of distant.c, obtained by multiplying the time delay by the speed of light. Acquiring measurements from the Goldstone-Canberra and Goldstone-Madrid baselines gives nearly perpendicular angular measurements in the plane-of-the-sky. Thus, ADOR data provides additional information which is not directly observable by range and doppler data. Also, ADOR data does not suffer from the well-known singularity affecting doppler data when the spacecraft is at or near 0° of geocentric declination [Ref. 3].

ADOR was used successfully on the Earth-1 encounter where the angular information was desirable duc to the low declination of the spacecraft for part of the approach trajectory [Ref. 2]. Although low declination was not a problem during Earth-2 encounter, it was still decided to incorporate ADOR data to improve accuracy. Originally, 21 points were scheduled. Of these, 19 were successful --10 points from the Goldstonc-Canberra baseline and 9 from the Goldstone-Madrid baseline. The ADOR campaign started on October 27, 1992 and ended on November 22, 1992. The predicted uncertainty for ΔDOR data was about 25 cm [J. Border, personal communication]. Operationally, ΔDOR data was weighted at 50 cm.

ORBIT DETERM1NATION STRATEGY

The navigation process begins with a precise nominal trajectory computed from integrating the equations of motion with a detailed force model. In the OD procedure, the different types of tracking data arc used to compare the actual trajectory with the nominal one. The residuals (ob.served values of the data minus the predicted values computed using the nominal trajectory) arc then used to adjust model parameters using the method of least squares to obtain a new, more accurate trajectory. Thus, the accuracy of the orbit determination procedure is dependent on the models used to propagate the spacecraft's orbit as well as the error sources affecting the data. The force models used to integrate the trajectory included the gravitational attractions of the Sun, Moon and the nine planets, solar radiation pressure, and spacecraft propulsive events. The gravitational effects included, in addition to the point mass accelerations of the Sun, Moon and the nine planets, the oblateness accelerations from the Earth and the Moon. Error sources affecting the data include path length delays duc to the troposphere and ionosphere and errors associated with DSN station locations.

In the OD solution process, many of the parameters in the force model and error sources arc estimated to provide a better orbit given the information provided by the data and a-priori uncertainties in those parameters. Along with the estimate, the least-squares formulat ion provides the formal computed uncertainty in the estimated parameters. If a parameter is not estimated, it can be "considered"; in other words, the a-priori uncertainty in these parameters is accounted for in the uncertainty of the solution, but it is not improved and does not affect the solution itself. For this portion of the mission, the standard estimated parameters included the state, solar radiation pressure, and all propulsive events.

Using a batch filter least-squares formulation, the state was estimated as the cartestian components of position and velocity at the initial epoch of the data arc. Solar radiation pressure was modelled using a flat plate representation of the spacecraft, and the estimated parameters in this model arc the specular and diffuse. reflectivity coefficients of the flat plate. Major propulsive events included four Trajectory Correction Maneuvers (TCMs). Impulsive AVS were estimated for each of these events. Smaller propulsive events included attitude update turns, propellant line flushings of the retro-propulsion module (RPM), and turns used for HGA anomaly recovery activities. Duc to the wide beamwidth of the LGA, the spacecraft did not have to be pointed directly at the Earth at all times, and attitude update-s occurred relatively infrequently. The size of these turns was under 15°, and all were estimated individually using impulsive ΔV's. Line flushings were used to clear the oxidant lines of the buildup of oxidants and occurred at roughly 23 day intervals. The flushings have known magnitudes and directions which were input prior to fitting the data. They gene-rally solved to within 1 mm/s of their a-priori values. HGA activities were larger turns (45° to 135°) used to point the spacecraft at an attitude which would either warm or cool the HGA, and were treated the same as altitude update turns. The a-priori uncertainties in the estimated parameters arc listed in Table. 1.

Since the strength of the S-band data is not sufficient to estimate the media calibration *errors,* these effects were considered. Included are separate parameters for the wet and dry components of the troposphere, and day and night components of the ionosphere at each station location. In addition, the station locations in the cylindrical coordinate system (longitude, height above the equator, and distance from the Earth's spin axis) were also considered in the filter. The a-priori uncertainties in all the consider parameters arc included in Table 1.

Finally, unique to this Earth encounter, atmospheric drdg effects also had to be taken into account because of the low flyby altitude. This was done two ways; the first and simpler method was to usc an impulsive AV in three Cartesian components occuring at the time of closest approach, and the second was to estimate drag directly using an atmospheric model. Although the first method was adequate for operational purposes, the second was needed in an attempt to pinpoint the contribution of atmospheric drag to the AV at perigee.

TABLE 1
A-PRIORI UNCERTAINTIES IN **THE ESTIMATED** AND CONSIDERED PARAMETERS

Estimated Parameters	A-priori Uncertainty (1 sigma)
State (Cartesian position and velocity)	10^8 km
Specular and diffuse radiation coefficients	10% of nominal solar radiation pressure value
TCMS	10% of nominal AV
Attitude update and HGA activity turns	2 mm/s, spherical
RPM's	1 mm/s along axial direction, 0.5 mm/s in the other two components, constrained
Considered Parameters	
Troposphere	40 cm wet, 10 cm dry
Ionosphere	75 cm day, 15 cm night
Station location	50 cm in spin radius, 6 m in z-height, 70 cm in longitude

The results of the orbit determination arc generally mapped to the B-plane of the target planet. The B-plane is defined as that plane intersecting the center of the target body and normal to the incoming asymptote of the spacecraft's hyperbolic trajectory. The axes in this coordinate system arc S, R, and 1, where S is parallel to the incoming asymptote, **R** is normal to S arid normal to the Earth ecliptic of 1950, and T is parallel to the Earth ecliptic of 1950 and normal to S such that **T = R x S.** This coordinate frame is illustrated in Figure 1. The vector **B** points from the origin of the coordinate system to the point where the incoming asymptote pierces the R-T plane (the R-plane). **B•R** and **B•T** arc the projections of **B** onto the **R** and **T** axes, respectively.

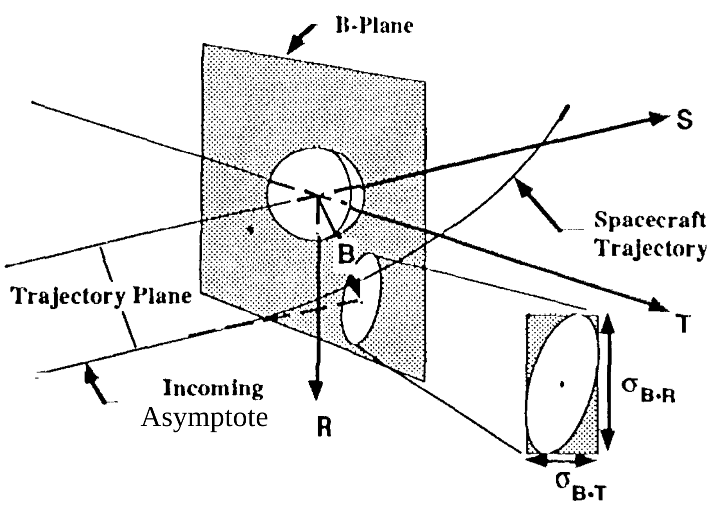

Figure 1 : I)-l'lane **Definition**

MANEUVERS

As **mentioned** earlier, four TCMs were performed to target the spacecraft from its post-Gaspra encounter trajectory to its final Earth flyby aimpoint (Figure 2). To lower the probability of impacting the Earth, the first two of these (TCMs 14 and 15) had "biased" aimpoints -- that is, they were deliberately targeted away from the optimum flyby point such that the probability of impacting the Earth was less than $1x10^{-6}$. TCM-16 was the first maneuver to place the spacecraft on its correct flyby trajectory, and an additional *maneuver* (TCM-17) was performed to clean up errors in TCM- 16. Figure 3 shows a plot of the change in the Earth B-plane caused by each of the TCMs. The characteristics and aimpoints in the Earth B-plane of the TCMs are shown in Table 2. The uncertainty in the aimpoint, caused by a combination of OD and maneuver execution errors, is also shown in the table.

TABLE 2
TRAJECTORY CORRECTION MANEUVERS PRIOR TO EARTH-2 FLYBY

TCM #	Date	Total ΔV Magnitude (m/s)	Selected Aimpoint and uncertainty (1 sigma)
14	Aug 4, 1992	21.27	B•R = 1662 ± 802 km B•T = -15,982 ± 1067 km TCA = 15:10:13 ± 376 sec
15	Oct. 9, 1992	0.72	B•R = 855.7 ± 131 km B•T = -12,430.9 ± 53 km TCA = 15:10:43 ± 7 sec
16	Nov. 13, 1992	0.89	B•R = 1097.4 ± 25 km B•T = -10,529.9 ± 24 km TCA = 15:09:25 ± 2 sec
17	Nov. 28, 1992	0.03	B•R = 1097.4 ± 4.2 km B•T = -10,529.9 ± 2.5 km TCA = 15:09:25.0 ± 0.1 sec

RESULTS

Table 3 shows the final OD solutions used to navigate the flyby. The solution is given in the Earth B-plane, along with the estimate for the Time of Closest Approach (TCA), given in UTC, on December 8, 1992, The uncertainties of the solution arc also provided Generally, a solution used to design a particular maneuver also provided the reconstruction of the previous maneuver.

An initially disconcerting result of the analysis was the large discrepancy between the TCM- 14 aimpoint and the actual trajectory as determined by OD#59, the post TCM- 14 solution (see Figure 4). The miss was about 1445 km in the B-plane -- well above the l-sigma dispersion for the TCM-14 delivery. Detailed analysis of the problem revealed the cause to be a combination of a timing error and an overburn in the thrusters.

The cumulative effect of these two maneuver implementation errors is shown in Figure 4. The aimpoint for TCM-14 is indicated, as well as the actual post-TCM-14 solution (labelled OD#59). The spacecraft timing error induced a 1600 km error in the B-plane, mainly in the B•R direction. Since Galileo is a spinning spacecraft, a timing error in the thrusters is equivalent to an angular offset in the position of the thrusters when they fire. The timing error was present in all subsequent maneuvers, but because those

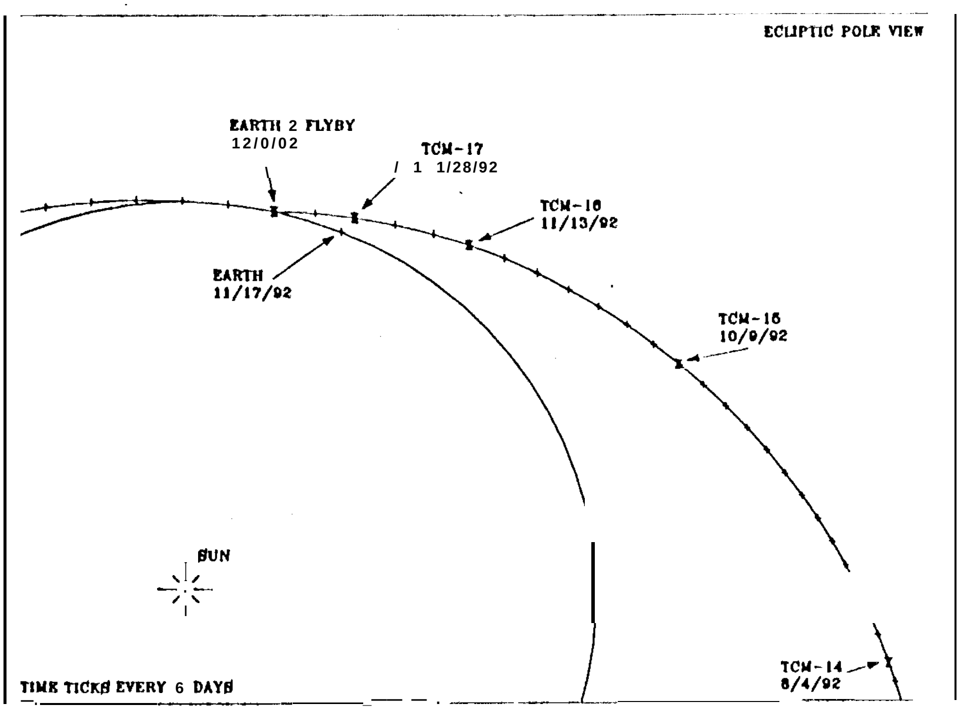

Figure 2: Overview of Galileo Trajectory from TCM-14 through the Earth Flyby

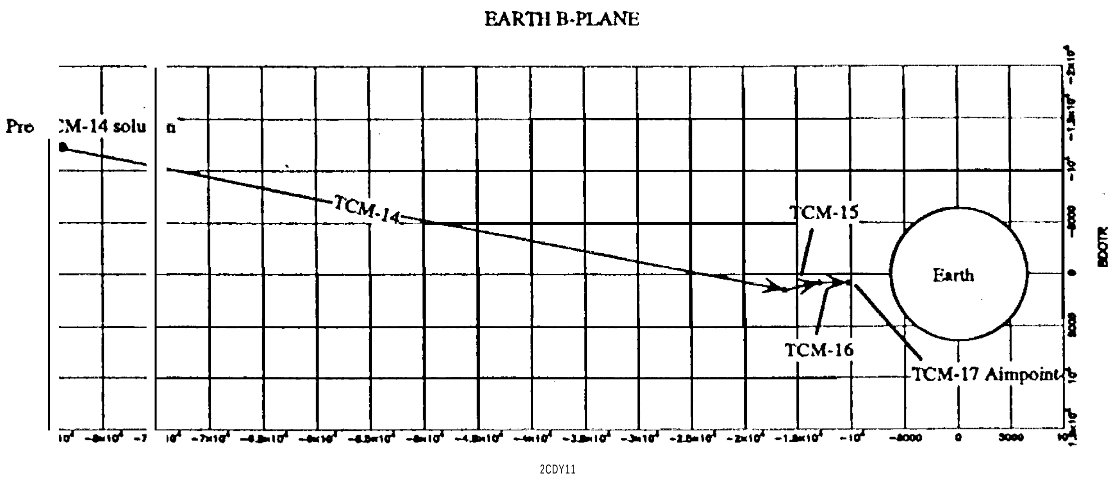

Figure 3: **Effect of the TCMs on the Trajectory in the Earth D-Plane**

maneuvers were fairly small, the effect of the error was well less than the uncertainty in the OD solution and was neglected.

The thrusters also overburned by 1.3%, and this caused an 850 km error in the B-plane, primarily in the B•T direction. As shown in the figure, the vector sum of the two B-plane offsets places the desired aimpoint within the 1 sigma uncertainly dispersion of OD#59.

OD#62, the solution used to design TCM-1 6, was the first to make extensive usc of ADOR data during the Earth approach. Seven points were fitted in the solution, of which four used the Goldstone-Madrid baseline and the other three were on the Goldstone-Canberra baseline. The effect of using ADOR data on the B-plane is shown in Figure 5, which compares solutions obtained with and without ADOR data. As can be seen in the figure, the B-plane coordinates computed using ΔDOR data was quite close to the radio only solution, but the dispersion ellipse was considerably smaller. The rms of the ADOR residuals was 20 cm, well within its predicted uncertainty.

TABLE 3
ORBIT DETERMINATION SOLUTIONS AND AIMPOINTS

Solutn #	Data Arc	OD Solution	Purpose
OD#56	515192- 7/16/92	B•R = -12296.5 ± 435 km B•T = -83521.8 ± 101 km TCA = 07:26:45 ± 15 sec	Design of TCM-14
OD#59	817192- 9/24/92	B•R = 361.2 ± 131 km B•T = -15354.7 ± 43 km TCA = 15:15:49 ± 5 sec	Reconstruction of TCM-14 Design of TCM-15
OD#62	8/8/92 - 1 1/2/92	B•R = 696.0 ± 23 km B•T = -12.,369.1 ± 12 km TCA= 15:10:41 ± 1 see	Reconstruction of TCM-15 Design of TCM-16
OD#64	10/1 5/92 - 11/20/92	B•R = 1082.4 ± 4.1 km B•T = -10531.7 ± 2.5 km TCA = 15:09 :27.8 ± 0.1 sec	Reconstruction of TCM- 16 Design of TCM-17
OD#68	11/29/92 - 12/10/92	B•R = 1096.2+ 0.02 km B•T = -10,529.2 ± 0,01 km TCA = 15:09 :24.9 ± 0,001 sec	Post flyby reconstruction

OD#62 was expected to fall within the TCM- 15 delivery, yet it did not. The stability of the different solutions and the size of the residuals lent confidence to the OD#62 solution. The question then arose as to why the solution was slightly over 1 sigma away from the aimpoint of TCM- 15. The timing and overburn error seen in TCM- 14 was *too* small in TCM-1 5 to explain the discrepancy. It had been determined previous to the TCM- 16 delivery that the.re was a discrepancy between the doppler and range solutions. In other words, a solution obtained from using only doppler data did not match a range only solution within their respective uncertainties. This discrepancy disappeared as the spacecraft approached Earth, and was completely gone by the time of the delivery. A very similiar situation occured at the corresponding time during the Earth-1 approach as well, although the discrepancy then was over 3 sigma. The error is possibly due to the geometry of the approach, but as of this writing, a definitive explanation has not been found.

OD#64 used all 19 (10 Goldstone-Madrid, 9 Goldstone-Canberra) available ADOR points in the solution. A comparison of the OD#64 and a corresponding radio-only B-plane solutions is shown in Figure 6. It can be seen that the addition of ADOR greatly improves knowledge of the trajectory. The dispersion ellipse of OD#64 is considerably smaller than what would have been obtained if ΔDOR were not available. This was due to the fact that the velocity imparted by TCM-16 was nearly perpendicular to the Earth-line direction.

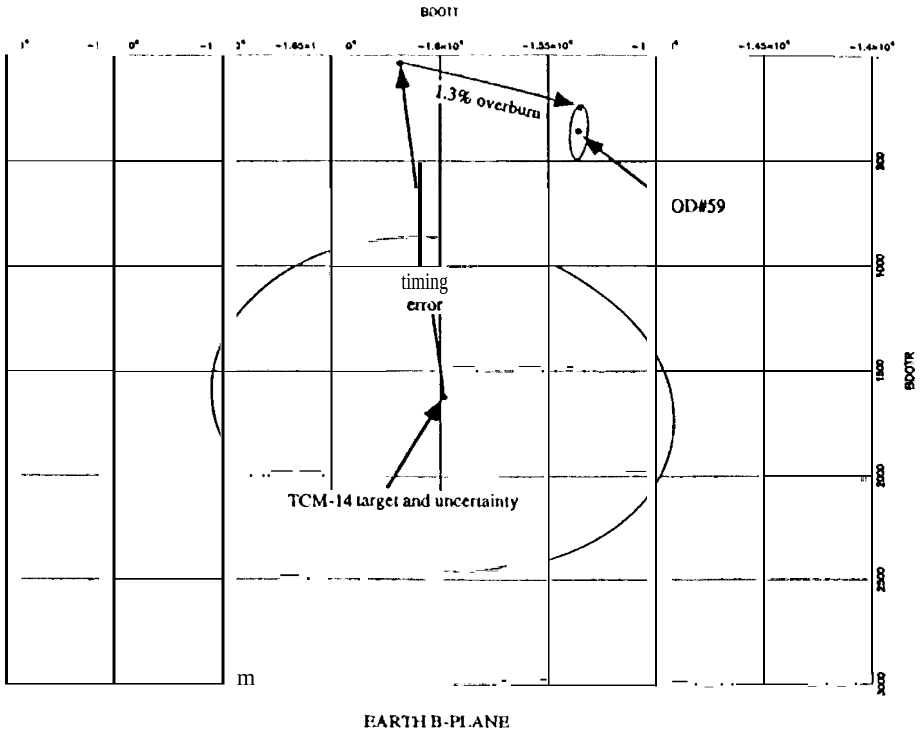

Figure 4: Pre-TCM-15 OD Results in the Earth B-Plane

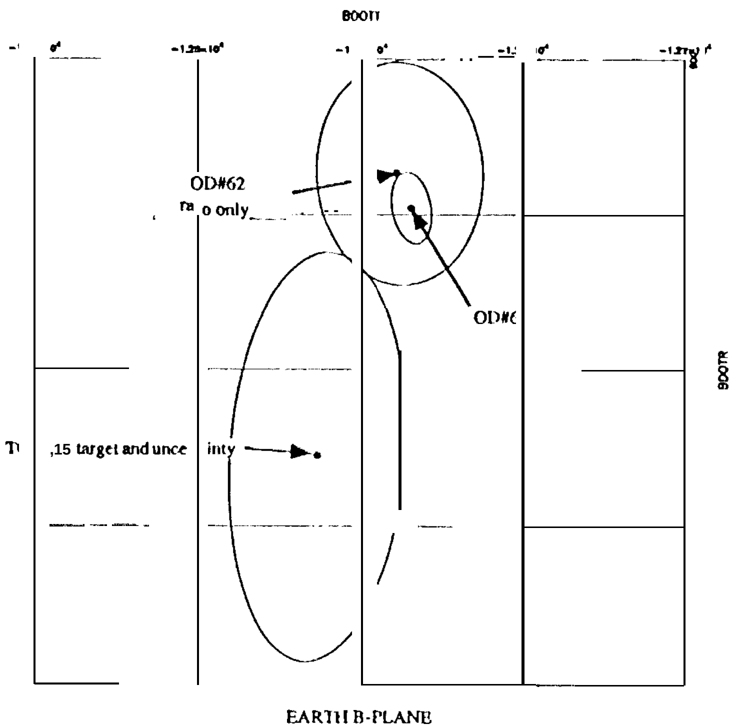

Figure 5: Pre-TCM-16 Results in the Earth **B-Plane**

Because doppler and range cannot directly sense this motion, the TCM- 16 AV was poorly solved for in the radio-only case. When ADOR data was added, a good solution was obtained for TCM-16 and its corresponding uncertainty was substantially reduced.

Figures 7,8, and 9 plot the doppler, range and ADOR residuals for OD#64. The rms of the ADOR residuals are about 21 cm, and no individual point is greater than 50 cm. The range and doppler residuals are also fairly fiat, showing the three data types to be in agreement (the 5-10 m biases in the range is primarily caused by solar plasma and station calibration errors). Based on this solution, TCM-17 was implemented to target the spacecraft back to its desired course.

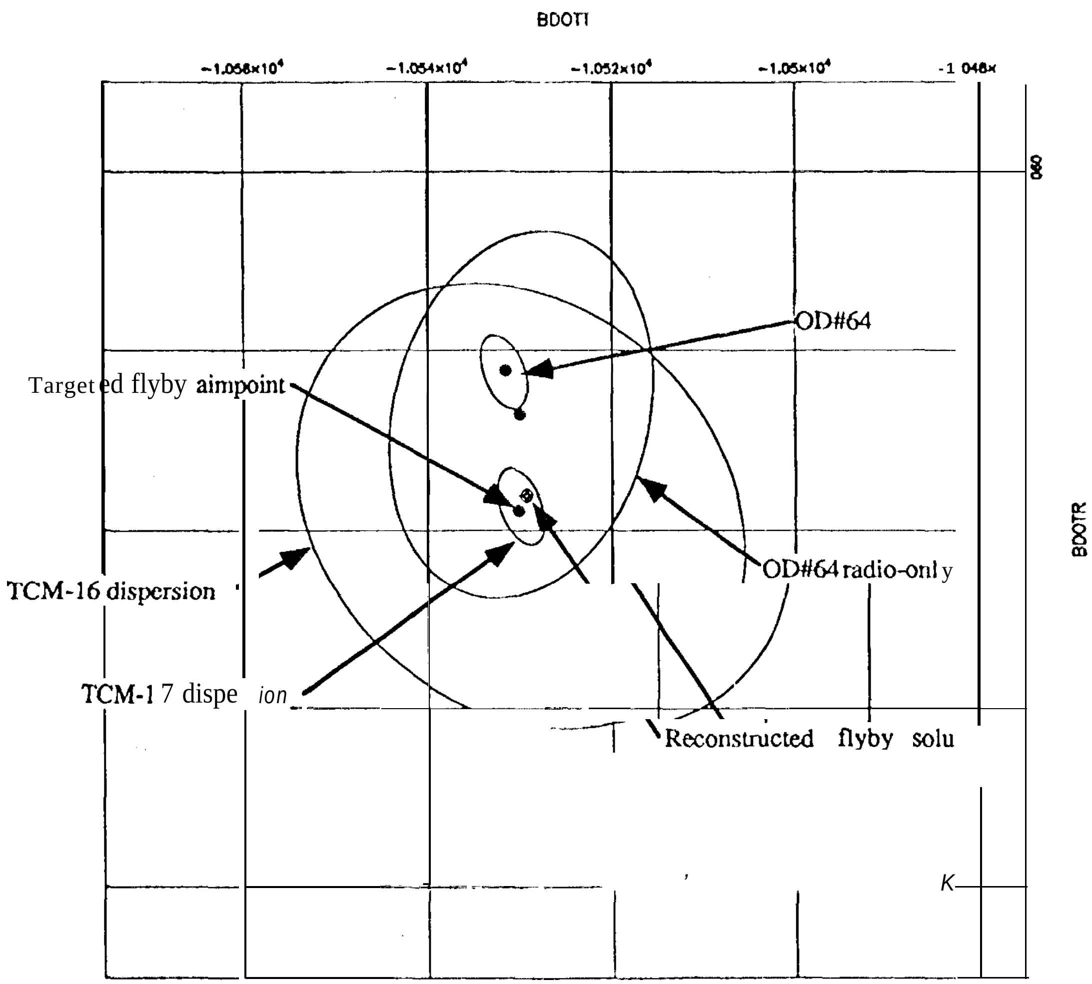

Figure 6: Final Flyby Results in the **Earth** B-Plane

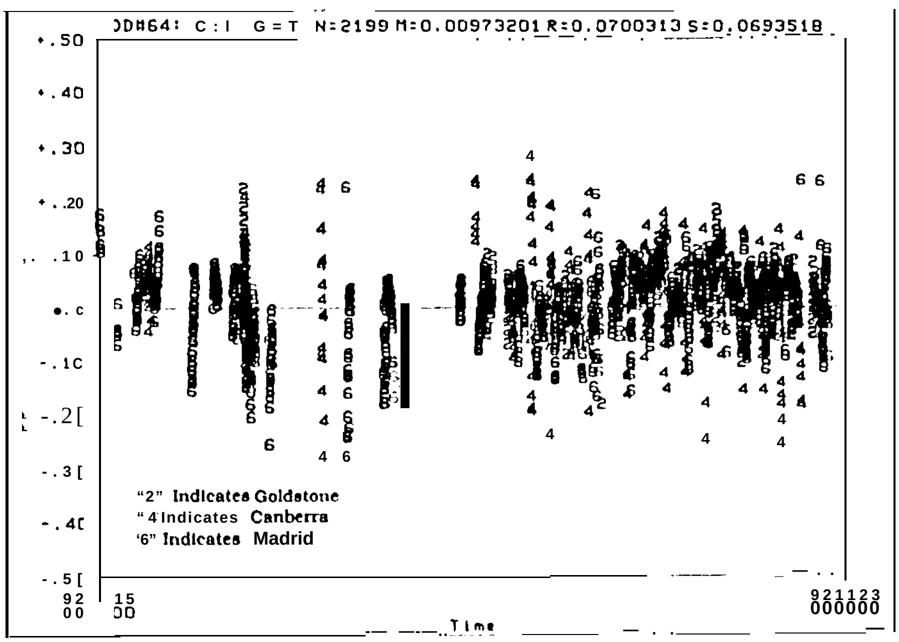

Figure 7: Doppler Residuals from **OD#64**

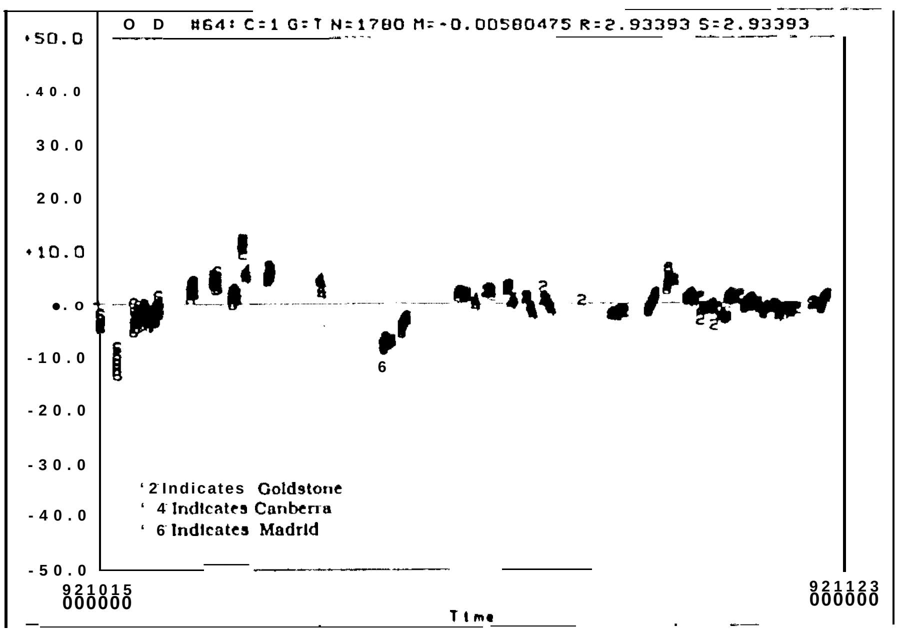

Figure 8: Range Residttakfrom OD#6d

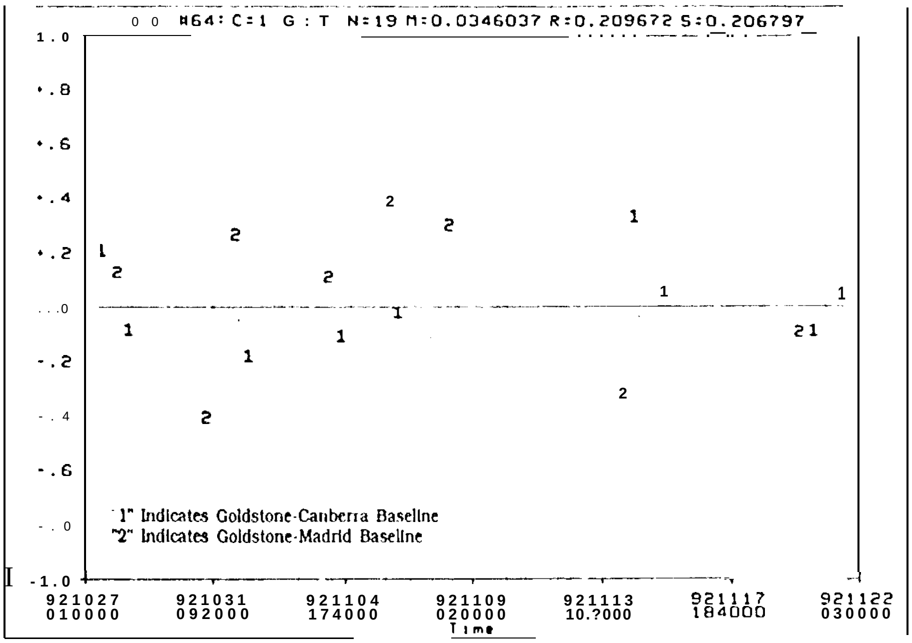

Figure 9: ΔDOR Residuals from OD#64

The post flyby reconstruction of the trajectory, OD#68, showed the spacecraft to have missed the desired aimpoint by only 1.4 km in the B-plane (0.7 km altitude error) and 0.1 se-c in the time of closest approach (see Figure 6). The data arc for OD#68 started on November 29, one day after TCM- 17, and ended on December 10, two days after the encounter. An important component of the reconstruction was to determine the effects of atmospheric dragon the spacecraft. This was necessary first to obtain an accurate trajectory for targeting to the asteroid Ma, and second to determine if an anomal y observed at the Earth- 1 encounter repeated itself*. For the former purpose, drag was estimated simply a.. an impulsive AV placed at the time of closest approach. Using this method, the total magnitude of the AV during encounter was found to be 5.94 mm/s. The majority of this was in the direction opposite to the velocity vector, with a magnitude and uncertainty of 5.89 ± 0.08 mm/s.

Although this result was adequate for operational purposes, it does not distinguish the causes of the AV at the encounter. Thus, analysis was undertaken to quantify the effects of drag during encounter. The simplest way to estimate drag directly is to introduce the drag equation into the force models. Then, the drag coefficient, CD, can be estimated, assuming fixed values for the atmospheric density, spacecraft area and mass. Using the computed C_D and values for the other parameters, the acceleration caused by drag can be found. The acceleration profile is then integrated to obtain a value for AV, which was found to be 6.0 ± 0.5 mm/s.

A separate study was also done to get a purely deterministic estimate for the AV caused by drag. The Jacchia-Roberts model was used to compute the density of the atmosphere using solar and geomagnetic activity for the time period surrounding the encounter. 'I'hen, using values for the spacecraft's drag coefficient and area, the trajectory was integrated and compared to a trajectory without any drag. The results

* The post-flyby reconstruction of the Earth- 1 flyby showed evidence of a velocity increase at closest approach of approximately 3 -4 mm/s. No cause has yet been determined for this anomalous frnding.

yielded a predicted AV due to drag of 6.21 ± 4.04 rends. 'I'his value is consistent with the values described above. The large uncertainty is dominated by the uncertaint y in the atmosphere at the altitudes through which the spacecraft flies. Without better knowledge of the atmosphere, it is unlikely that the drag can be quantified with any more accuracy.

CONCLUSION

Galileo was successfully navigated through the Froth-2 encounter using a combination of doppler, range and ADOR data. The final Earth-2 encounter was only 1,4 km in the B-plane and 0.1 seconds from the planned target. In addition, the high accuracy with which the spacecraft was placed on its optimum flyby trajectory resulted in the cancellation of TCM-1 8, the maneuver scheduled two weeks after the Earth-2 encounter to clean up errors in TCM- 17 and the flyby.

The reconstruction of the first Earth-2 targeting maneuver showed for the first time evidence of a slight timing error in the thruster firings. Future maneuvers of this type will be designed to alleviate this error. The remaining three Earth-2 flyby maneuvers performed nominally. The three methods used to account for atmospheric drag result in similar results. Nevertheless, a AV anomaly of the type seen at Earth- 1, if it occurred, could not be resolved duc to the uncertainties in the drag for the Earth-2 flyby.

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